

foregoing amendments and the following remarks is respectfully requested.

In response to the Examiner's position as set forth in paragraph 1 on page 2 of the Office Action, applicants note that a Secrecy Order has been imposed on the application in response to a request filed February 14, 1991.

By this Amendment the specification has been amended so as to provide appropriate headings and also to effect editorial changes therein; however, no new matter has been introduced into the specification by virtue of the present amendment.

Additionally, by this Amendment a new Abstract of the Disclosure has been submitted in response to the Examiner's requirement as set forth in paragraph 4 on page 4 of the Office Action.

With regard to the Examiner's position as set forth in paragraph 5 of the Office Action, applicants attach hereto a Supplemental Declaration identifying the instant application by serial number and filing date.

By this Amendment the claims have been amended so as to correct the improper multiple dependency thereof; therefore, it is respectfully requested that the Examiner reconsider the objection as set forth in paragraph 6 of the Office Action and withdraw said rejection.

The rejection of the claims under 35 USC 112, second

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paragraph, as allegedly being indefinite, as applicable to the presently amended claims, is respectfully traversed.

By this Amendment the claims have been amended taking into account the Examiner's helpful suggestions as contained in paragraph 7 of the Office Action. Consequently, applicants respectfully submit that the claims, as amended, are in full compliance with the requirements of 35 USC 112, and particularly point out and distinctly claim the subject matter which the applicants regard as their invention. Therefore, it is respectfully requested that the Examiner reconsider the rejection of the claims under 35 USC 112, second paragraph, and withdraw said rejection.

The rejections of the claims under 35 USC 103 as unpatentable over Nissan Jidosha KK, Hydran Products Ltd. and Kishi, et al for the reasons set forth in paragraphs 9 and 10 of the Office Action, as applicable to the presently amended claims, are respectfully traversed.

The Examiner contends that the Nissan Jidosha KK '846 Japanese reference proposes a ramjet engine comprising a cruising propulsion unit 5, a combustion chamber 4, a gas ejection nozzle, an air duct, a transverse portion, and passages. The Examiner notes that the Japanese '846 reference does not disclose a cruising propulsion unit with a liquid feeding fuel; however, Hydran Products Ltd. teaches a cruising propulsion unit with a liquid feeding fuel. Consequently, the Examiner concludes that the applicants are

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merely substituting propulsion of one type for another which is commonly known in the art. Therefore, the Examiner contends that to employ the teaching of the Hydran Products Ltd. on the ramjet of the Nissan Jidosha KK so as to provide a different type of propulsion fuel would be obvious to one of ordinary skill in the art. Applicant respectfully submits that the Examiner's contentions and conclusions, as applicable to the presently amended claims, are without merit as the following analysis reveals.

At the outset, applicants note that in securing a translation of the above-identified application the translator mistakenly translated the French word "fluide" to "liquid" when, in fact, in the instant situation, the French word "fluide" should have been translated to "gaseous" as apparent from a review of the last two lines of page 14 and the first six lines of page 15 wherein it is explained that solid propellant block 27 generates reduction gases. By this Amendment the text has been corrected accordingly. Consequently, by virtue of this correction in the text, which finds a clear basis in the original text, applicants respectfully submit that a detailed discussion of the alleged applicability of the Hydran Products Ltd. patent is deemed unnecessary with regard to its applicability to the subject matter of the present invention since the fuel in question is a gaseous not liquid fuel.

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Turning to the Japanese '846 reference applicants respectfully submit that presuming arguendo that the reference does in fact propose the features outlined at the bottom of page 6 of the Office Action, the reference is also clearly deficient with respect to the provision of a tubular element made of a composite material composed of resistant fibers coated with a polymerized synthetic resin and also is deficient with respect to the provision of an insert means for enabling an attachment of an end of at least one air duct to the tubular element in a vicinity of the combustion chamber as well as the provision of a pyrotechnic fuse means for cutting openings into the combustion chamber through which openings the air ducts communicate with the combustion chamber. Furthermore, the Japanese '846 reference is also deficient with respect to the remotest teaching or suggestion of insert means shaped so as to act as cutting knives to cut a wall of the tubular element upon an actuation of the pyrotechnic fuse means as also required by claim 1 and the claims dependent thereon.

A review of Hydran Products Ltd. as well as Kishi, et al reveals that such references do not cure the above noted deficiencies of the Japanese '846 reference. Consequently, even if, for some reason, one skilled in the art were to attempt to modify the Japanese '846 reference in the manner suggested by the Examiner in the Office Action, the so modified construction would in no way resemble the subject

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matter recited in amended claim 1 and claims 3-9, 11, 12 and 14 which depend therefrom and which recite additional perfecting features of the present invention.

Accordingly, in view of the foregoing remarks, applicants respectfully submit that the claims, as amended, clearly patentable define over any attempted combination of the Japanese '846 reference, Hydran Products Ltd. and Kishi, et al. Therefore, it is respectfully requested that the Examiner reconsider the rejection of the claims for the reasons set forth in paragraphs 9 and 10 of the Office Action and withdraw said rejection.

In accordance with the provisions of 37 C.F.R. 1.56, applicants attach hereto a copy of a Search Report Issued in connection with a corresponding U.K. application along with a copy of British 2218494A, 2,104,628A, 1,421,719, 1,417,350 and United States Patent 4,332,637.

While it is believed that the instant Amendment places the application in condition for allowance, should the Examiner have any further comments or suggestions, it is respectfully requested that the Examiner telephone the undersigned in order to expeditiously resolve any outstanding issues.

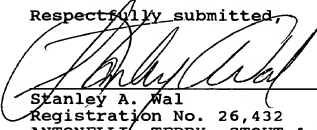
In view of the foregoing amendments and remarks, reconsideration of this application is respectfully requested and an early and favorable action upon all the claims is earnestly solicited.

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To the extent necessary, applicant petitions for an extension of time under 37 C.F.R. 1.136. Please charge any shortage in fees due in connection with the filing of this paper, including extension of time fees, to the Deposit Account of Antonelli, Terry, Stout & Kraus, Account No. 01-2135 (659.27786X00) and please credit any overpayment of fees to such deposit account.

Respectfully submitted,



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Patents Act 1977
Examiner's report to the Comptroller under
Section 17 (The Search Report)

Application number

8922949

#50
5/9/91
4/24/91

Relevant Technical fields

(i) UK CI (Edition K) F3A; F1J

(ii) Int CI (Edition 5) F02K 7/10, 7/12, 7/14, 7/16, 7/18, 7/20

Databases (see over)

(i) UK Patent Office

(ii) ON-LINE: DERWENT'S WPI, CLAIMS

Search Examiner

J S BOOTH

Date of Search

25 JULY 1990

Documents considered relevant following a search in respect of claims

All

Category (see over)	Identity of document and relevant passages	Relevant to claim(s)
X, E	GB 2,218,494A (SOC. NATIONALE)	1, 12 at least
X	GB 2,104,628A (M-B-B-)	1, 12, 13 at least
X	GB 1,421,719 (WILKINSON)	1 at least
X	GB 1,417,350 (HERCULES)	1 and 6 at least
X	US 4,332,631 (HERCULES)	1, 12 at least

(12) UK Patent Application

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F02K 9/26 // F02K 7/18 9/18 9/28 9/30 9/36

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F3A A10H1 A10H2B A10L1 A10L5 A10N A10R2
A10R3

(56) Documents cited

None

(58) Field of search

UK CL (Edition J) F3A
INT CL⁴ F02K

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(54) Propulsion unit

(57) A propulsion unit 1 for the acceleration of a self-propelled vehicle, especially for a ram-jet engine with an integral accelerator and a gas generator 2, includes a propellant block 10 secured laterally to the body of the propulsion unit by combustion inhibitor 13 and having an aspect ratio (L/D) of between 2.5 and 6. The propellant block has one axial duct 11 and at least 6 peripheral ducts 12, the ducts each terminating in a diverging part 16, and, on its upstream face, is equipped with a fitting 14 secured to the body of the propulsion unit. The axial duct 11 and the peripheral ducts 12 open into a free space 20 for balancing pressures in the ducts via orifices 15 made in the fitting.

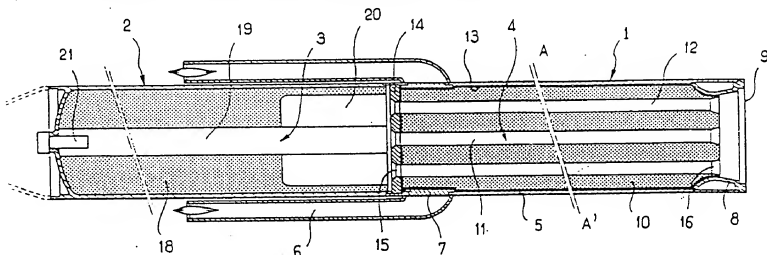


FIG. 1

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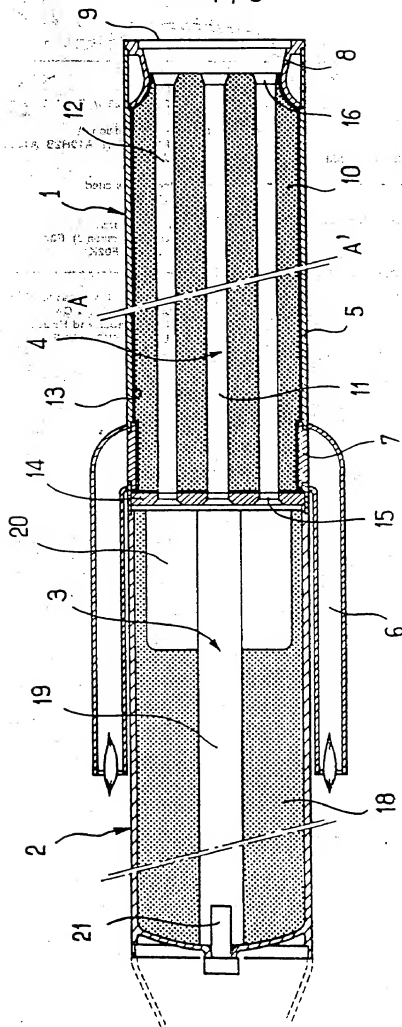
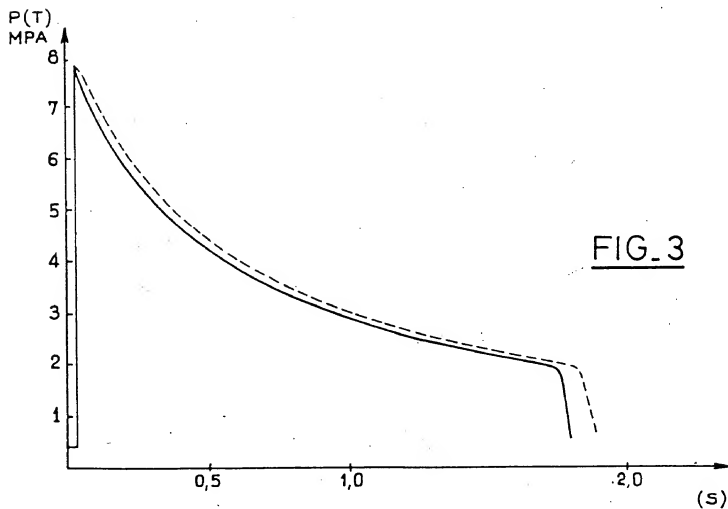
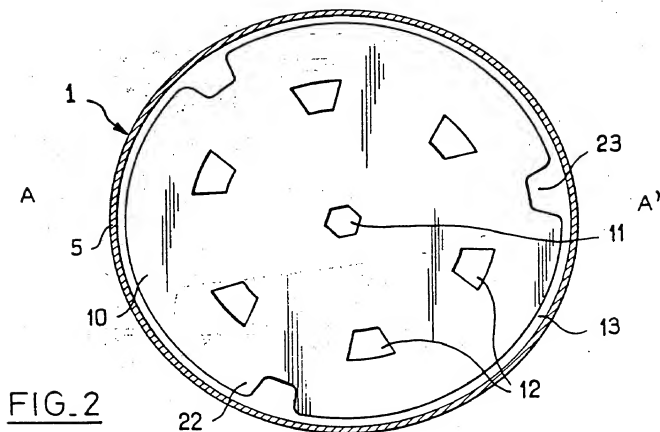
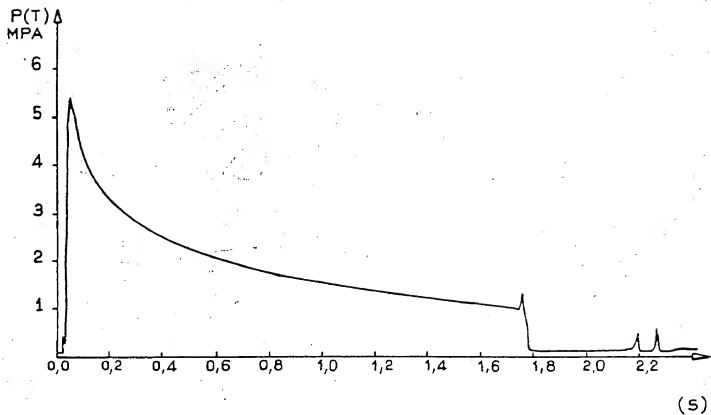
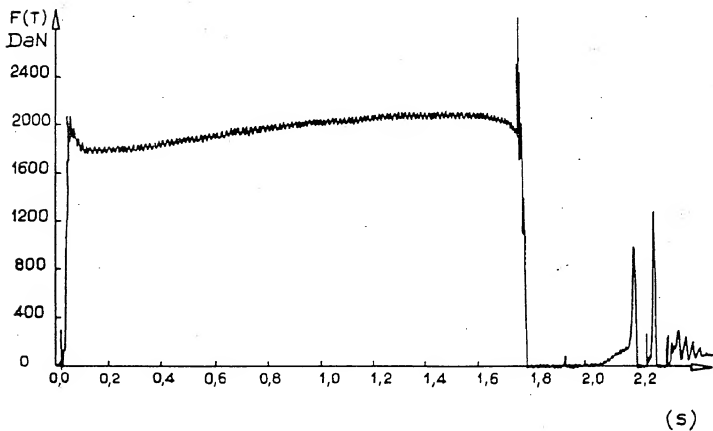


FIG. 1



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FIG. 4FIG. 5

Propulsion unit of low aspect ratio

The invention relates to a propulsion unit for the acceleration of a self-propelled vehicle, such as a missile or a rocket, loaded with a propellant block having several ducts. The use of a single-duct nozzleless propulsion unit is known particularly for the acceleration of ram-jet engines with an integral accelerator, such as those described in U.S. Patent 3,535,881. The relevant propulsion units have an aspect ratio below 6. By the aspect ratio of a cylinder is meant the ratio L/D of its length to its diameter.

The combustion rates of composite propellant blocks are too low to allow a sonic section to be established in a duct of an aspect ratio below 6. Since the aspect ratio of the duct is linked to the aspect ratio of the block, it is therefore difficult to use propellant blocks of an aspect ratio below 6. Now in some uses, size constraints prevent blocks of an aspect ratio higher than 6 from being used.

For example, the combustion chamber of a kerosene ram-jet engine of a diameter of 350 mm requires only a length of 800 mm. Its aspect ratio is therefore merely of the order of 2.3. However, on the other hand, it offers sufficient volume to accommodate in it the propellant necessary for accelerating the

vehicle so driven up to the so-called transition speed, at which the ram-jet engine is to take over from the accelerator. There is therefore no need to lengthen it.

5 In the present state of the art, it will not have a jetless accelerator incorporated in it, but a conventional accelerator equipped with an ejectable nozzle seated in the nozzle of the ram-jet engine, because it is of a cross-section much smaller than
10 that of the latter, for reasons of the operating pressures, and jetisoned at the cutoff of the accelerator; thus possibly presenting a danger especially when the vehicle is launched from an aircraft.

15 According to the present invention there is provided an acceleration propulsion unit comprising a propellant block secured laterally to a propulsion unit body by combustion inhibitor and having an aspect ratio between 2.5 and 6, an axial duct and at least
20 six substantially identical peripheral ducts in the body, a fitting secured to the body at the upstream face of the propellant block, and a free space defined in the body on the side of the fitting remote from the propellant block, at one end the axial and peripheral
25 ducts opening into the free space through orifices in the fitting, and the ducts each terminating at the opposite end in a divergent part.

The acceleration propulsion unit according to the invention operates without a jet nozzle, even
30 though it is equipped with a propellant block of an aspect ratio of between 2.5 and 6.

Each of the ducts taken individually has a sufficient aspect ratio to ensure that the velocity of the gases reaches sound velocity in a section near the
35 downstream end of the duct, and downstream of this

section posses a divergent part making it possible to expand the gasses with acceptable efficiency. These ducts open onto the rear face of the block, in such a way that the resultant of the thrust vectors is borne along the axis of the vehicle to be accelerated.

Because all the ducts open into the free space provided upstream of the block, this space performs the function of pressure-balancing chamber, and the change of pressure as a function of time in each duct is such that there is substantial equality of the pressures at each moment. To minimize the combustion residues, the peripheral ducts are arranged on one or two concentric circles and number 6 or 12 respectively. According to a preferred feature of the invention, the block has seven ducts, namely 1 axial duct and 6 peripheral ducts.

According to another preferred feature, the 6 peripheral ducts are parallel to the axial duct.

Furthermore, the peripheral ducts of such a charge are at a distance from the outer surface of the propellant block corresponding to half the thickness separating the ducts from one another.

The central duct can be of circular or hexagonal cross-section.

The peripheral ducts located on the concentric circle or circles can be of a cross-section suitable, on the one hand, for the type of propellant, used, to limit the interactions between the various ducts at the moment of combustion, and, on the other hand, for the potential incidence of the combustion residues. The cross-section of these ducts can therefore, be a circle, a regular hexagon, a pentagon or a trapezium, with the knowledge that the large base of the trapezium can be curved or straight.

Advantageously, the peripheral ducts have a cross-

section substantially in the form of an isosceles trapezium. The trapezoidal form is useful because it reduces the combustion residues to a minimum and maintains efficient combustion and efficient gas flow for the longest possible time.

The pressure difference Δp prevailing between the two end faces of the propellant block is increased as a result of the high aspect ratio of the ducts. Now the propellant block thus perforated is more fragile than a single-duct block and consequently than a front combustion block.

It is well known to an average person skilled in the art to secure the propellant block laterally to the body of the propulsion unit by means of a combustion inhibitor. It has been found that this fastening is insufficient for a multiply perforated propellant block. The propellant block of the propulsion unit according to the invention has, on its upstream face, a fitting secured to the body of the propulsion unit, this fitting providing a free space between the upstream face of the propellant block and the bottom of the body of the propulsion unit, the axial duct and the peripheral ducts opening into this free space via orifices made in the fitting.

According to a preferred feature of the invention, the said fitting consists of a perforated plate, of which the orifices of the same number as the ducts of the propellant block are in line with the peripheral ducts and the axial duct, and is fastened to the upstream face of the propellant block by means of the combustion inhibitor.

The free space thus provided between the fitting fastened to the upstream face of the propellant block and the bottom of the propulsion unit forms a chamber for balancing the internal pressure,

which reduces the pressure differences liable to arise during combustion. This precaution is justified because, if the propellant thickness separating two ducts and becoming increasingly small in proportion as combustion proceeds were subjected to a pressure difference, at the end of combustion this difference would give rise to the premature rupture of the propellant not yet burnt, with a risk of ejection of propellant fragments, and would cause an irregularity at the end of combustion and thereby a considerable loss of efficiency. This pressure-balancing chamber located at the upstream end of the acceleration propellant can be arranged downstream of the gas generator of a ram-jet engine.

The upstream fitting of the propellant has orifices which allow the combustion gases, that is to say the pressure, to be distributed among the ducts by way of the balancing chamber. These orifices can have the exact initial form of the ducts in the propellant or a completely different form. For example, the ducts can be of trapezoidal cross-section and the orifices in the fitting can be cylindrical or shaped as truncated cones.

The advantage of orifices of cylindrical or truncated cone form is that this fitting can be used again for the second propulsion phase, in the case of a ram-jet engine arrangement, in order to distribute the jets of combustible gases; the fitting is then used as a diffuser/injector.

The fitting is fastened to the body of the propulsion unit and incorporates an inhibition, particularly by the return of the inhibitor securing the propellant to the tube of the propulsion unit. The inhibitor serving for protecting this fitting can

be arranged on either face of the latter, but will preferably be arranged on the rear face of the fitting, that is to say between the propellant block and the fitting itself, to prevent it from being heated to excess.

According to a preferred feature, the propulsion unit according to the invention forms the acceleration stage of the ram-jet engine. In this case, the downstream end of the propellant block is located in the region of the nozzle of the ram-jet engine and preferably in the region of the divergent part of this nozzle, such a configuration making it possible to accommodate more propellant in the body of the propulsion unit.

Each duct terminates in a divergent part, in order to improve the expansion of the gases, and where the acceleration block of a ram-jet engine is concerned this widened part is positioned in the region of the nozzle of the ram-jet engine.

The combustion of the propellant takes place in parallel layers, and at the end of combustion non-burned remains, called combustion residues, remain on the periphery of the inner inhibitor fixed to the structure of the propulsion unit; the central residues are ejected.

According to a first alternative version, these residues can be partially eliminated by using a combustion-inhibiting material which penetrates into the outer cylindrical surface of the propellant block in at least two zones located in the region of these residues, thus forming longitudinal ribs affording the advantage of reducing the risks that combustion instabilities will occur during the functioning of the ram-jet engine.

According to another alternative version,

these residues are eliminated completely and rapidly by using a twin-composition block, the propellant of higher combustion rate being arranged in the zones located in the region of the external combustion residues.

According to a preferred feature of the invention, the propellant used is a composite propellant with a hydroxytelechelic polybutadiene binder and contains up to 5% by mass of ferrocenic catalysts.

The invention is described below by way of example with reference to Figures 1, 2 and 3.

Figure 1 shows a propulsion unit according to the invention used as an integral accelerator of a ram-jet engine.

Figure 2 shows cross-section AA' through the acceleration propulsion unit according to Figure 1.

Figure 3 shows the theoretical pressure curves existing in the peripheral ducts and in the central duct of the propellant block of the acceleration propulsion unit as a function of the combustion time.

Figure 4 shows the pressure curve as a function of the combustion time.

Figure 5 shows the thrust curve as a function of the combustion time.

Referring to Figure 1, a ram-jet engine with an integral accelerator consists of an acceleration propulsion unit (1) according to the invention and of a gas generator (2).

The combustion chamber of the gas generator is called a primary chamber (3). The combustion chamber of the acceleration propulsion unit, initially occupied by a propellant block, is called the secondary chamber (4).

The body (5) of the acceleration propulsion unit (1) is equipped, at its upstream end, with air inflow orifices (6) closed by means of caps (7) during the acceleration phase and, at its downstream end, with a nozzle (8) of large cross-section, intended for expanding the combustion gas of the ram-jet engine and initially closed by means of cap (9).

The secondary chamber of the ram-jet engine is equipped with an acceleration propellant block (10) with a PBHT binder, multiply perforated with a longitudinal central duct (11) and 6 and a longitudinal peripheral ducts (12). This block (10) has a diameter of 191 mm for a length of 500 mm, that is to say an aspect ratio of 2.61.

This multiply perforated propellant block (10) is secured laterally to the body of the propulsion unit (5) by means of a combustion inhibitor (13). The upstream face of this block has a plate (14) perforated with frustoconical orifices (15) in the same number as and in line with the peripheral ducts (12) and central duct (11) of the propellant block (10), to which is fastened by means of a combustion inhibitor.

The downstream face of the propellant block (10) is located in the region of the divergent part of the nozzle (8), and the peripheral ducts (12) and the central duct (11) terminate in divergent end parts (16).

The gas generator is equipped with a suboxygenated propellant block generating reduction gases (18), which is equipped with a central duct (19) and which is secured to a primary chamber (3).

The suboxygenated propellant block does not occupy the entire space of the primary combustion chamber: a free space (20) is provided between the

perforated plate (14) and the downstream end of the block (18), to form a pressure balancing chamber.

An igniter (21) is located at the upstream end of the central duct of the suboxygenated propellant block (18).

It should be noted that the igniter (21) could also have been placed in the region of the free space (20).

According to Figure 2, the multiply perforated propellant block (10) is secured laterally to the body (5) of the acceleration propulsion unit (1) by means of a combustion-inhibiting material (13). This block is perforated with a hexagonal central duct (11) of a height of 42 mm and with 6 peripheral ducts (12) arranged on a circle concentric relative to the central duct (11).

The 6 peripheral ducts (12) have a cross-section substantially in the form of an isosceles trapezium, likewise of a height of 42 mm, and they are arranged in such a way that their flanks are parallel two by two.

The combustion-inhibiting material (13) penetrates into the outer cylindrical surface of the propellant block (10) in three zones located in the region of the combustion residues (22), thus having three longitudinal ribs (23) parallel to the ducts.

During the launching of the vehicle equipped with such a ram-jet engine with an integral accelerator, the combustion gases of the igniter (21) pass through the central duct (19) of the gas-generating block (18) and flow through the orifices in the plate (14) into the ducts of the multiply perforated block (10) of the accelerator which they ignite. The pressure in the entire volume not occupied by propellant (ducts of the accelerator block, balancing

chamber and duct of the gas-generating block) rises quickly and causes the gap (9) to be ejected, thus allowing the ejection of the combustion gases of the accelerator and causing the acceleration of the vehicle.

Since the combustion gases of the igniter (21) pass through the central duct of the gas-generating block (18), they also initiate the pyrolysis of the latter. The pyrolysis gases produced during the combustion period of the accelerator are ejected together with the combustion gases of the latter and are lost in terms of the subsequent functioning of the ram-jet engine, but they contribute slightly to the acceleration of the vehicle.

The multiply perforated propellant block (10) burn within approximately 1.8 seconds, and the ejection of the combustion gases accelerates the vehicle up to a sufficient speed to allow the ram-jet engine to take over from the accelerator.

Combustion takes place in parallel layers. As can be seen in Figure 3, the curve of pressure as a function of time is decreasing and the pressure prevailing in the peripheral ducts (12) is substantially equal to that prevailing in the region of the central duct (11), thus reducing the risk of premature rupture of the block. Furthermore, the presence of the ribs (23) in the region of the combustion residues allows combustion to stop sharply. Once the combustion of the multiply perforated propellant block (10) has ended, the pressure in the combustion chamber decreases, thus causing the ejection of gaps (7) via the secondary combustion chamber (4) and the massive inflow of air through the orifices (6).

The pyrolysis gases of the generator ignite spontaneously with the air under the pressure and

temperature conditions of the latter. The mixture of air and of pyrolysis gases in combustion experiences a considerable increase in temperature and is ejected through the nozzle of large cross-section (8); this initiates functioning in the ram-jet engine mode.

Figures 4 and 5, illustrating the bench-test launching curves for a multiply perforated accelerator charge, show that the ignition of the charge takes place after 20 milliseconds and that the combustion lasts for 1.8 seconds. At this time, a pressure peak attributable to the ejection of the central residues is seen on the curves.

The shapes of the pressure and thrust curves are comparable to those of known single-duct jetless charges, namely a decreasing pressure curve and a slightly increasing thrust curve.

CLAIMS:

1. An acceleration propulsion unit comprising a propellant block secured laterally to a propulsion unit body by combustion inhibitor and having an aspect ratio between 2.5 and 6, an axial duct and at least
5 six substantially identical peripheral ducts in the body a fitting secured to the body at the upstream face of the propellant block, and a free space defined in the body on the side of the fitting remote from the propellant block, at one end the axial and peripheral
10 ducts opening into the free space through orifices in the fitting, and the ducts each terminating at the opposite end in a divergent part.

2. A propulsion unit according to Claim 1,
15 wherein the propellant block has only six peripheral ducts.

3. A propulsion unit according to Claim 1 or 2,
20 wherein the peripheral ducts are parallel to the axial duct.

4. A propulsion unit according to Claim 1, 2 or 3, wherein the peripheral ducts have a cross-section substantially in the form of an isosceles trapezium.

5. A propulsion unit according to one of the preceding claims, wherein the unit forms the acceleration stage of the ram-jet engine with an integral accelerator.

6. A propulsion unit according to Claim 5,
30 characterized in that the downstream end of the propellant block is located in the region of the divergent part of the nozzle of the ram-jet engine.

7. A propulsion unit according to one of the preceding claims, wherein the propellant of the propellant block contains a ferrocenic catalyst.

5 8. A propulsion unit according to one of the preceding claims, wherein said fitting consists of a perforated plate and the orifices therein are aligned with the ducts.

10 9. A propulsion unit according to Claim 8, wherein the perforated plate is fastened to the upstream face of the propellant block by combustion inhibitor.

15 10. A propulsion unit according to anyone of the preceding claims, wherein an ignitor is located at least partially in the region of the free space.

20 11. A propulsion unit according to Claims 5, 8 or 9, wherein the orifices in the plate serve as injection orifices for reduction gases of emanating from an upstream charge.

25 12. A propulsion unit according to anyone of the preceding claims, wherein the combustion inhibitor penetrates into the outer surface of the propellant block in at least two zones located in the region of expected external combustion residues.

30 13. A propulsion unit according to anyone of Claims 1 to 11, wherein the propellant block is of bicomposition, the propellant of higher combustion rate being arranged in zones located in the region of expected external combustion residues.

14. A propulsion unit according to anyone of Claims 1 to 13, wherein the propellant block is a composite propellant with a PBHT binder.

5 15. An acceleration propulsion unit substantially as herein described with reference to the accompanying drawings.

10



(12) **UK Patent** (19) **GB** (11) **2 104 628 B**

(54) Title of invention

Combustion chamber closure means for an air inlet aperture of a rocket combustion chamber

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F3A 10R3

(56) Documents cited
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GB 1348894
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GB 1299851
US 3901028
US 3724216
US 3535881

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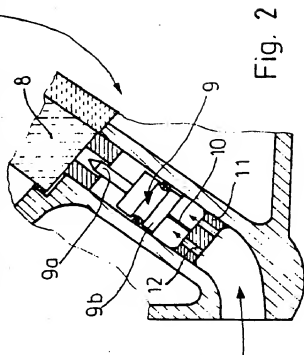
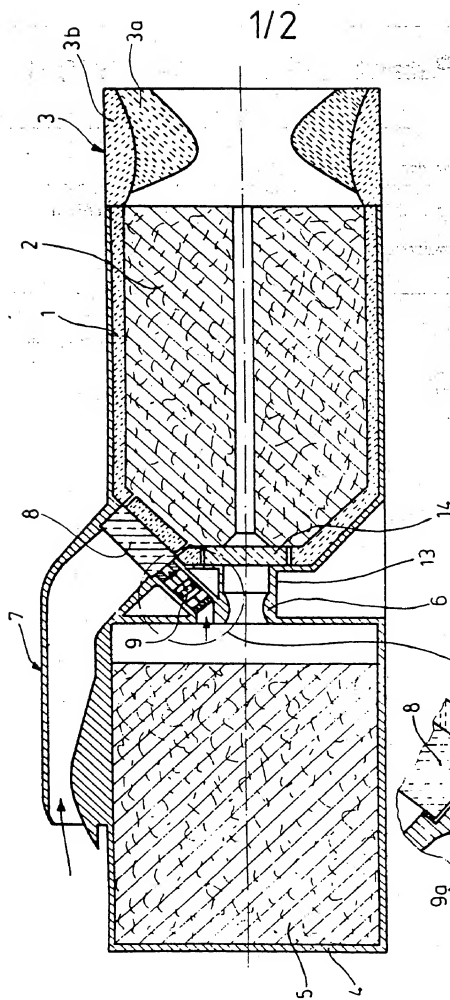
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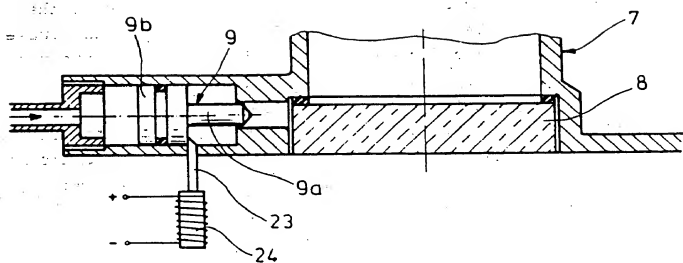
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F1J
F3A
F1D
F4P

(60) Derived from Application No.
8101381 under Section 15(4) of
the Patents Act 1977



2/2

Fig. 3



Combustion chamber closure means for an air inlet
aperture of a rocket combustion chamber.

5 This invention relates to a destructible closure for
an inlet duct communicating with a combustion chamber
especially one in a rocket ramjet propulsion engine
and a striker device for destroying the cover at the
end of the launching phase and beginning of the cruise
10 phase, the striker being driven into contact with the
cover by gas pressure generated in the engine.

In a ramjet rocket engine, as disclosed for example
in U.S. Patent 3 901 028, a solid rocket propellant is
provided for the launching phase which brings the
15 projectile up to the high flight velocity required
for operation of the ramjet to provide sufficient air
at adequate pressure for the cruising phase. During
the latter phase oxygen from the air is caused to
react in the combustion chamber with gases which have
20 a high proportion of propellant and which are generated
by a solid fuel.

The combustion chamber is followed by a convergent-
divergent thrust nozzle serving to convert the pressure
25 energy generated in the combustion chamber into
reaction force. In order to ensure that the air inlet
apertures leading into the combustion chamber remain
closed during the launching phase, the inlets are pro-
vided with closures of frangible material which, at
30 the end of the launching phase and at the commencement
of the cruising phase, are destroyed by forces applied

thereto, so that the air inlet apertures are then open and unobstructed. The prevailing high ram air pressure causes the fragments of the cover to be forced out through the combustion chamber and the thrust
5 nozzle. The cover may be destroyed by a pyrotechnic charge or by a mechanical striker device using a prestressed spring.

One of the difficulties arising with such a system is
10 that of ensuring that the cover will be destroyed to open the air inlet passages at the desired instant. It is also important that the cover should be caused to break-up into small fragments, so as not to cause damage to the projectile or internal parts of the
15 propulsion unit or other projectiles in the vicinity.

This invention seeks to provide an air inlet closure made of a material which, on the destruction of the cover, will not be detrimental to the safety of the
20 engine unit or other projectiles used at the same time, and wherein the destruction of the cover will take place at a moment calculated in advance or under pre-determined operating conditions.

25 According to one aspect of the present invention there are provided closure means for an air inlet aperture feeding a combustion chamber of an engine, the closure means comprising a closure member made of an easily destructible material, and a striker assembly for
30 breaking same, the closure member being principally

of a material which is internally prestressed and the striker assembly incorporating means responsive to gas pressure acting thereon for urging the striker towards the said closure member and means for restraining the
5 striker against displacement by the said gas pressure until the gas pressure acting thereon reaches a predetermined value or a release signal is generated.

In one embodiment the said means for restraining the
10 striker element is a frangible link which ruptures when the said predetermined pressure value is reached. Preferably the said frangible link is secured at one end to the striker element and at the other end to a housing or a part integral with the housing for the
15 striker and is so oriented as to be placed under tension by the force acting on the striker assembly due to the said gas pressure.

In an alternative embodiment the said restraining means
20 comprise a movable blocking member having means for displacing it from a blocking position retaining the striker element to a position where movement of the striker is unobstructed. Conveniently the said means for displacing the movable blocking member include a
25 electromagnet device for causing displacement of the blocking member upon application of an electrical control signal.

The easily destructible material of the cover may be
30 an internally prestressed glass, although other mat-

erials, such as ceramics may alternatively be employed.

The present invention also comprehends a rocket ram-jet propulsion engine with a launching propulsive charge
5 in a combustion chamber having an air inlet aperture and closure means as hereinabove defined, and a cruise propulsive charge, the striker assembly of the closure means being effective to destroy the closure upon
10 ignition of the cruise propulsive charge. Preferably the gas pressure for driving the striker is derived from the combustion of fuel in a precombustion chamber having an outlet leading to the said combustion chamber.

Two embodiments of the invention will now be more particularly described by way of example, with reference
15 to the accompanying drawings, in which:

Figure 1 is a longitudinal section through a ramjet and rocket propulsion unit having a frangible cover
20 formed as a first embodiment of the present invention;

Figure 2 is an enlarged view of a detail of the embodiment illustrated in Figure 1; and

25 Figure 3 illustrates a second embodiment of the invention.

As shown in Figure 1, a rocket and ramjet propulsion unit has a common combustion chamber 1 for the launch-
30 ing and for the cruising phase, this containing a

launching propulsive charge 2 in which the oxygen required for complete combustion is bound chemically. The combustion chamber 1 has at the rear end a thrust nozzle 3 with a throat part 3a for the launching phase and a throat part 3b for the cruising phase. 5 The thrust nozzle part 3a is continuously eroded during the launching phase. The forward end of the ramjet has a pre-combustion chamber 4 containing a cruising propulsion charge 5 which, in the course of its consumption during the cruising phase, generates 10 gases with a high fuel content which are passed through a nozzle 6 to the common chamber 1, where additional oxygen from the air for the purpose of stoichiometric combustion is supplied. The air is supplied through 15 one or more air inlet ducts 7 which are closed by a closure 8 until the end of the launching phase.

The closure 8 comprises a prestressed material, preferably internally prestressed glass such as ordinary 20 commercial thermally prestressed glass which was and is widely used for motor vehicle windscreens and which disintegrates into very small fragments when subjected to mechanical overloading due to the high compressive stresses generated in the surface of the 25 glass by the prestressing. The prestressing is usually imparted by thermal treatment. Chemical treatment can also be used, sodium ions in the surface of the glass, for example, being replaced by calcium ions. 30 Glass is not the only material that can be thermally or chemically prestressed and processes can likewise

be applied to ceramic materials such as aluminium oxide.

5 The closure 8 is destroyed when required by a mechanical striker device 9 illustrated in more detail in Figure 2 and comprising a bolt 9a tapering to a point at the front and a plunger 9b which is retained in the position shown by a restraining element 10 connected to a retaining device 11 which is connected or
10 integral with the housing and which has apertures 12. The gas pressure generated in the combustion chamber passes through the bores and acts on a piston 9. When the pressure level reaches a predetermined value in the pre-combustion chamber 4, the restraining element 10 breaks and the bolt 9a is driven, by the
15 gas pressure acting on the piston 9, to impact against the closure 8, which disintegrates into tiny fragments to be expelled harmlessly through the thrust nozzle 3. The strength of the restraining element 10 is therefore adapted to the required pressure. The plate 14
20 sealing the passage 13, and preferably of the same material as the closure 8, is also destroyed when the reference pressure is reached.

25 In the second embodiment illustrated in Figure 3 a bolt 23 is provided for the purpose of restraining the striker 9 in the initial position and this is released at the correct moment by an electromagnet device 24, whereby the striker 9 can then be driven,
30 urged by pressure from combustion chamber 4 acting on the piston 9b, into contact with the closure 8. Because release of the striker 9a is controlled by the electrical signal fed to the electromagnet device 24 the precise instant when the closure 8 is destroyed
5 can be controlled or predetermined.

CLAIMS

1. Closure means for an air inlet aperture feeding a combustion chamber of an engine, the closure means comprising a closure member made of an easily destructible material, and a striker assembly for breaking same, the closure member being principally of a material which is internally prestressed and the striker assembly incorporating means responsive to gas pressure acting thereon for urging the striker towards the said closure member and means for restraining the striker against displacement by the said gas pressure until the gas pressure acting thereon reaches a predetermined value or a release signal is generated.

2. Closure means as claimed in Claim 1, in which the said means for restraining the striker element is a frangible link which ruptures when the said predetermined pressure value is reached.

3. Closure means as claimed in Claim 2, in which the said frangible link is secured at one end to the striker element and at the other end to a housing or a part integral with the housing for the striker and is so oriented as to be placed under tension by the force acting on the striker assembly due to the said gas pressure.

4. Closure means as claimed in Claim 1, in which the said restraining means comprise a movable blocking

member having means for displacing it from a blocking position retaining the striker element to a position where movement of the striker is unobstructed.

5 5. Closure means as claimed in Claim 4, in which the said means for displacing the movable blocking member include an electromagnet device for causing displacement of the blocking member upon application of an electrical control signal.

10 6. Closure means as claimed in any of Claims 1 to 5, in which the easily destructible material is internally prestressed glass.

15 7. Closure means substantially as described herein with reference to Figures 1 and 2 or Figure 3 of the accompanying drawings.

20 8. A rocket ramjet propulsion engine with a launching propulsive charge in a combustion chamber having an air inlet aperture and closure means according to any preceding claim, and a cruise propulsive charge, the striker assembly of the closure means being effective to destroy the closure upon ignition
25 of the cruise propulsive charge.

 9. A rocket ramjet propulsion engine as claimed in Claim 8, in which the gas pressure for driving the striker element is derived from the combustion of fuel
30 in a precombustion chamber having an outlet leading to the said combustion chamber.

The text of the specification has been reproduced by photocopying the applicants original typescript. It may contain a few amendments which are difficult to read. The original typescript containing these amendments may be inspected on the premises of the Patent Office

1 421 719

- (21) Application No. 37258/74 (22) Filed 23 Aug. 1974 (19)
 (31) Convention Application No. 392 380 (32) Filed 28 Aug. 1973 in
 (33) United States of America (US)
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 (52) Index at acceptance
 F3A 10R3
 FIJ 2B4



(54) IMPROVEMENTS IN OR RELATING TO RAMJET POWERED MISSILES

(71) I, DAVID BRUCE WILKINSON, a citizen of the United States of America, of 1489 Lemcke Road, Xenia, Ohio 45385, United States of America, do hereby declare the invention, for which I pray that a patent may be granted to me and the method by which it is to be performed, to be particularly described in and by the following statement:—

The present invention relates to ramjet powered missiles and, more particularly, to such missiles which are fuelled with solid fuel.

Solid ramjet powered missiles have several advantages over liquid fuel ramjets. The fuel is safer because fuel leakage cannot occur. Also, burning is more stable since the hot fuel grain tends to stabilize combustion in the primary zone and fuel utilization automatically responds to changes in altitude and velocity. In prior art solid fuel ramjet powered missiles, the engine air inlet is placed forward of the forward end of the fuel grain with bypass channels being provided to channel air to the secondary combustor as described in United States Patent No. 2,799,987. Since the primary air flow is normally only a small percentage of the total flow, large ducts are required to supply the major portion of the air to the secondary combustor. Also, since the rate of burning of the fuel grain is a function of the exposed area, either the fuel grain has to be specially designed to provide a uniform burning rate or an uneven burning rate with poor thrust control will result.

The present invention consists in a solid fuel ramjet powered missile having a payload section, a ramjet section including a fuel grain with an air passage therethrough and means for igniting the fuel grain, a secondary combustor positioned at the aft end of the fuel grain, an air inlet having its lip positioned near the aft end of said fuel grain, means within said inlet for directing a major portion of the air, entering the inlet, into the secondary combustor, and a channel adjacent the fuel grain, extending from said inlet to a position adjacent the forward end of the fuel grain, for providing primary air to the forward end of the fuel grain.

Positioning the air inlet or inlets near the aft end of the fuel grain produces increased missile stability since the static aerodynamic stability of the missile is reduced as the inlets are moved forward. With the air inlet(s), the large air flow is directly into the secondary combustor which reduces pipe flow losses. Also, since the primary air flow is normally much less than the secondary air flow to the secondary combustor, a smaller size channel or duct can be used for primary air flow thus reducing the overall weight of the missile. A flow control valve may be provided in the primary air flow channel so that the burning rate of the fuel grain can be controlled.

In order that the invention may be more readily understood, reference will now be made to the accompanying drawings, in which:—

Figure 1 is a schematic partially cut away sectional view of a solid fuel ramjet powered missile according to the invention, and

Figure 2 is a sectional view of the missile taken along the line 2—2 of Figure 1.

Reference is now made to Figure 1 of the drawings which shows a solid fuel ramjet missile 10 having a conventional payload, such as a target acquisition section 12, an ordnance section 14, a flight control section 16 and a ramjet section 18, with stabilizing fins 19 located at the aft end of the missile.

The ramjet section 18 has a solid fuel grain 20 positioned within chamber 22 with the exit nozzle 24 at the aft end of the secondary combustor 26. The solid fuel grain may be any conventional air burning fuel grain such as magnesium and aluminum with a rubber binder. For some applications, a small amount of oxidizer may be added with the amount added being insufficient to maintain combustion in the absence of air. The ram air inlet 28 has its lip 29 positioned near the trailing end of the fuel grain with about 85% of the ram air being directed into the secondary combustor 26 by flow vanes 30. A conventional boundary layer bleed is provided at 25. A small portion of the ram air, for example about 15%, is bled off by flow director 32 and passed along channel 34 to the forward end of the fuel grain to provide

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primary air for burning the fuel grain. Primary combustion occurs under fuel rich conditions in the longitudinal channel 35 through the center of the fuel grain. Fuel rich products of primary combustion are further burned in the secondary combustor 26.

Since the fuel system of the solid propellant burner is easily throttleable by throttling the primary air flow, a valve 37 is provided to control the flow of primary air into the fuel grain channel 35. The valve 37 may be positioned by a hydraulic, pneumatic or screw jack drive system 38 which may be controlled either by a command signal from the flight control section or in response to a preset program. For example, the missile could be programmed to provide a constant thrust as the area of the fuel grain changes with burning.

A conventional igniter 41, such as a pyrotechnic material ignited with squib, initiates the burning of the fuel grain in a conventional manner upon command from the flight control section, through lines 43, which may also carry the control signal to fins 19.

Although not shown, the missile can be adapted for use with a rocket-ramjet engine by providing a rocket fuel grain in chamber 26 and configuring nozzle 24 for hybrid operation in a conventional manner.

While only one inlet scoop and one primary air channel are shown, more than one may be provided if desired.

There is thus provided a solid fuel ramjet

powered missile with increased static stability, lower inlet air pipe flow losses and a controlled burning rate.

WHAT I CLAIM IS:—

1. A solid fuel ramjet powered missile having a payload section, a ramjet section including a fuel grain with an air passage therethrough and means for igniting the fuel grain, a secondary combustor positioned at the aft end of the fuel grain, an air inlet having its lip positioned near the aft end of said fuel grain, means within said inlet for directing a major portion of the air, entering the inlet, into the secondary combustor, and a channel adjacent the fuel grain, extending from said inlet to a position adjacent the forward end of the fuel grain, for providing primary air to the forward end of the fuel grain.

2. A missile as claimed in claim 1, including a flight control section, and means responsive to a signal from the flight control section for controlling the flow of primary air into the fuel grain.

3. A solid fuel ramjet powered missile, constructed and adapted to operate substantially as hereinbefore described with reference to the accompanying drawings.

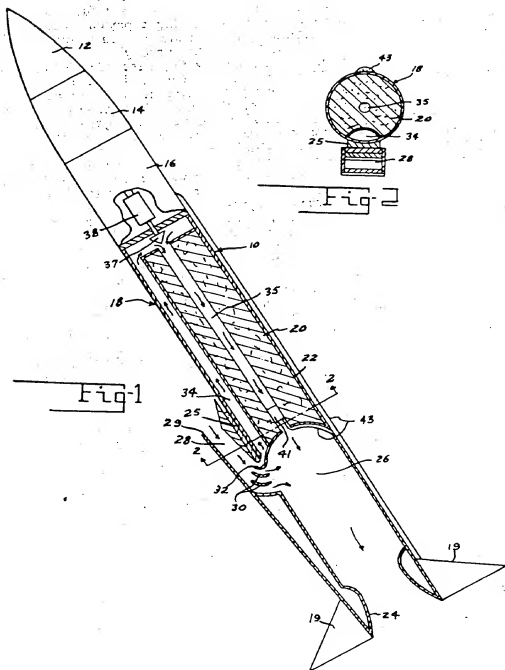
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COMPLETE SPECIFICATION

1 SHEET

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the Original on a reduced scale



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(72) Inventors RICHARD PENCE CARTER and
 FREDERICK JOSEPH POLICELLI

(54) FILAMENT WOUND VESSEL AND METHOD OF MANUFACTURE

(71) We, HERCULES INCORPORATED, a corporation organized under the laws of the State of Delaware, one of the United States of America, of 910 Market Street, City of Wilmington, State of Delaware, United States of America, do hereby declare the invention, for which we pray that a patent may be granted to us, and the method by which it is to be performed, to be particularly described in and by the following statement:—

This invention relates to filament wound vessels suitable for instance for rocket motor cases, and to their method of manufacture.

A rocket motor case commonly comprises a vessel closed at one end and open at the other to which a nozzle is attached. A gas, generated within the motor at elevated pressure, exits through the nozzle and imparts a thrust in the opposite direction. This vessel or rocket chamber must have sufficient integrity to withstand the internal gas pressure. The present invention relates to a design and method of manufacture of vessel which offers several economic and performance advantages as a rocket case over existing types. The invention is more specifically directed to filament wound vessels.

Small vessels used as rocket cases are usually made with a full diameter opening at one end or the other, the reason for this being to facilitate loading or casting of the propellant charge into its casing. Metals such as high strength steel have been used for the most part to make these cases. However, composite materials, usually fiberglass bonded together with a resin, have been used to some extent for the small cases and to a considerably wider extent for large casings. The advantages of the latter for small and large casings reside in cost saving generally.

Composite rocket cases have been made with full openings. However, rather heavy or bulky and expensive metal closures are

employed, which may be either screwed, bolted, riveted, pinned or keyed together. Other techniques provide for bonded together composite type rocket cases. With these techniques a short section of one part is smaller in diameter than the other, the two parts slide together and are held in place with an adhesive. A considerable amount of time, material, and effort are needed to join these parts. Similarly, pressure bottles and pressure tanks have been manufactured as composite structures and, accordingly, this invention is directed to filament wound cylindrical vessels generally which are adapted to withstand internal pressure from whatever source.

Generally described, the present invention contemplates a filament wound cylindrical vessel adapted to withstand internal pressure comprising an outer shell, and an inner shell slidably disposed within the outer shell, said inner shell having a hoop strength to axial strength ratio lower than that of said outer shell whereby the said inner shell expands more than the outer shell, upon application of internal pressure, to form a strong frictional bond between the shells. As a preferred embodiment thereof the inner shell is loaded with rocket propellant and the outer shell includes a rocket nozzle essentially to provide a rocket motor.

Moreover, the present invention includes the method for manufacture of the aforementioned pressure vessel and rocket motor. Embodiments of the invention have been chosen for purposes of illustration and description and are shown in the accompanying drawings wherein:

Fig. 1 is a longitudinal, part sectional and part elevational view of a gas-generating device in the form of a cylindrical rocket motor illustrating the application of the invention; and

Fig. 2 is a longitudinal, part sectional and part elevational view of a cylindrical vessel adapted to withstand internal pressure

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illustrating the application of the invention.

Referring now to Fig. 1, a rocket motor 2 has an outer shell 4 having the fore end with a full diameter opening 6 and having the aft end terminating as a rocket nozzle 8 integrally wound with the outer shell 4. An inner shell 10 has the fore end thereof forming a closure with an adapter 12 for receiving an igniter (not shown) and has a propellant 14 disposed therein. The inner shell 10 with its components 12 and 14 is slidably disposed within the outer shell 4 by engaging the full diameter aft end 16 of the inner shell 10 with the full diameter fore end 6 of the outer shell 4, and sliding the inner shell 10 thereinto until its aft end engages the necked down portion 18 of the rocket nozzle 8. This places the fore end closure of the inner shell to the inside of the fore end full opening of the outer shell. It is well known that the stress in the circumferential mode of a rocket casing or other vessel under pressure is twice that of the longitudinal stress. The winding of a

composite pressure vessel can be performed to a proper advantage by building into the design the required balanced strength, i.e., twice as strong in girth as in length. This has been disclosed by R. E. Young in U.S. Patent No. 3,083,864.

The strength of a given winding in the girth or hoop direction is a function of the \sin^2 of the winding angle, whereas the longitudinal or axial strength is a function of the \cos^2 . Therefore, the designer of a filament wound composite structure can vary the strength in a given direction by changing the winding angle or use relative amounts of layers wound at different angles. The \sin^2 of the angle $54\frac{1}{2}^\circ$ is twice that of its \cos^2 . Therefore the strength of a vessel wound with this angle would be balanced against internally developed pressure. A winding angle of 45° has equal strength in each direction. Therefore the strength of a combination of one part of a 45° winding and one-half an amount of a 90° winding would be balanced against internal pressure.

$$\text{Hoop Strength: } \sin^2 45^\circ \text{ plus } \frac{1}{2} \sin^2 90^\circ = 1.00$$

$$\text{Axial Strength: } \cos^2 45^\circ \text{ plus } \frac{1}{2} \cos^2 90^\circ = .50$$

In practice for rocket motors a ratio of hoop to axial strength of less than 2 is employed to overcome the cylinder to dome discontinuity.

With reference to the foregoing and to Fig. 1, Owens-Corning S/904 glass fiber with an epoxy resin was used, and 288 ends to the inch per layer was applied. The inner shell 10 was wound with seven such layers at 28° plus a layer about half as thick at 90° producing a hoop to axial strength ratio of 0.38 to 1. The outer shell 4 was wound with six layers at 38° plus six at 90° . The strength ratio of this shell was 2.22 to 1. Rocket motors made with the aforementioned windings were loaded with conventional propellant and successfully fired.

It will be seen therefore that this invention provides for the manufacture of the inner shell of the two-part rocket chamber with a lower hoop to axial strength ratio than the outer part. Upon pressurization the inner shell expands more readily than the outer. For optimum design the coefficient of friction for the materials used can be obtained and this, calculated with the area of contact involved, gives the strength ratio imbalance needed. Taking these factors into account no adhesive is required to bond the inner and outer shells.

Referring now to Figure 2, a cylindrical vessel 20 adapted to withstand internal pressure has an outer shell 22 with one end thereof with a full diameter opening 24. The other end thereof forms a closure 26 having an adapter 28 for receiving a valve body or

conduit (not shown). An inner shell 30 has one end thereof forming a closure 32 in combination with a plugged pole piece 34 adjacent the opening 24 of the outer shell 22. The other end of the inner shell 30 has a full diameter opening 36 adjacent the closure 26 of the outer shell 22. The inner shell 30 is slidably disposed within the outer shell 22 in the same manner as described for the rocket motor of Figure 1. The winding of the shells and determination of desired hoop to axial strength and frictional coefficient are likewise determined.

Accordingly the present invention provides a method for the manufacture of a filament wound vessel able to withstand internal pressure, which comprises (a) forming an outer shell having an open-ended cylindrical part, by winding filamentary material to provide a predetermined ratio of hoop strength to axial strength; (b) forming a separate inner shell also having an open-ended cylindrical part, by winding filamentary material to provide a ratio of hoop strength to axial strength less than that of the outer shell, the inner shell having an external diameter less than the internal diameter of said outer shell such as to provide a sliding fit of the one in the other; and (c) forming the vessel by sliding the said inner shell, open end first, into the cylindrical part of the outer shell so that when internal pressure is applied to the vessel, the inner shell will expand and develop a frictional bond between the shells.

The invention further provides a filament wound vessel comprising: (a) a filament

wound outer shell having an open-ended cylindrical part at one end thereof and a closure or restricted opening at the other end thereof, and (b) a filament wound inner shell disposed telescopically within the outer shell and having an external diameter representing a sliding fit within the internal diameter of the cylindrical part of the outer shell, said inner shell having a closure or restricted opening at the end thereof nearer the open end of the outer shell and an opening at its other end, said inner shell having a hoop strength to axial strength ratio lower than that of said outer shell so that when internal pressure is applied to the vessel, said inner shell will expand more than the outer shell and develop a frictional bond between the shells.

It will be appreciated that although glass filaments and epoxy resin have been used in the examples of this invention that other filamentary material may be used including carbon filaments and boron filaments with epoxy or other resin systems.

The usual tool or mandrel for winding composite pressure vessels consists of one which will dissolve, disintegrate or come apart after the item is cured. The present invention makes possible the use of a simple, solid or permanent mandrel. Two half-shells are fabricated on such a suitable mandrel and, when cured, are parted and the two halves slipped off over the mandrel ends. Using such a solid and very accurate mandrel makes possible an accurate inside diameter tolerance. The outside diameter will vary somewhat depending on materials and conditions but the variation usually will be within 5 per cent of the shell wall thickness. The inner shell, therefore, must be machined or ground in a centerless grinder on the outside to a very accurate tolerance to produce more or less close sliding fit of the two parts.

Another advantage of this invention is such that the outer shell can be made longer than the inner shell. Thus, after assembly, the extra length of the outer shell affords a convenient attachment stub or skirt to fasten, for example, a rocket to a warhead. This invention also applies to composite pressure vessels other than rockets, such as, gas storage cylinders, bottles and tanks. In this case, the slightly extended outer shell makes a convenient base for upright support as particularly depicted in Fig. 2.

Still other advantages of this invention are that it makes possible composite rockets or pressure vessels which are much simpler to make, load and assemble than those of the prior art. Generally, the prior art has employed elaborate means of injecting an adhesive into a carefully machined cavity or the like to bond the two halves of a composite vessel together. However, this and

similar methods are time consuming and expensive to perform. The present invention reduces the closure effort to its ultimate simplicity, i.e., pressing together two open ended shells. For permanence in the unpresurized state a thin film of adhesive may be wiped on the outside of the inner shell before assembly. Therefore, the advantage of large composite rocket cases, namely, lighter weight, less expense and no missile hazard, can be extended to smaller rocket units and pressure vessels.

WHAT WE CLAIM IS:—

1. Method for the manufacture of a filament wound vessel able to withstand internal pressure, which comprises (a) forming an outer shell having an open-ended cylindrical part, by winding filamentary material to provide a predetermined ratio of hoop strength to axial strength; (b) forming a separate inner shell also having an open-ended cylindrical part, by winding filamentary material to provide a ratio of hoop strength to axial strength less than that of the outer shell, the inner shell having an external diameter less than the internal diameter of said outer shell such as to provide a sliding fit of the one in the other; and (c) forming the vessel by sliding the said inner shell, open end first, into the cylindrical part of the outer shell so that when internal pressure is applied to the vessel, the inner shell will expand and develop a frictional bond between the shells.

2. The method as claimed in Claim 1, in which a thin film of adhesive is applied to the inner shell prior to engagement of the inner and outer shells, whereby their engagement is maintained in the absence of applied internal pressure.

3. A method of manufacturing a filament wound vessel substantially as described with reference to the accompanying drawings.

4. A filament wound vessel comprising: (a) a filament wound outer shell having an open-ended cylindrical part at one end thereof and a closure or restricted opening at the other end thereof, and (b) a filament wound inner shell disposed telescopically within the outer shell and having an external diameter representing a sliding fit within the internal diameter of the cylindrical part of the outer shell, said inner shell having a closure or restricted opening at the end thereof nearer the open end of the outer shell and an opening at its other end, said inner shell having a hoop strength to axial strength ratio lower than that of said outer shell so that when internal pressure is applied to the vessel, said inner shell will expand more than the outer shell and develop a frictional bond between the shells.

5. A filament wound vessel according to Claim 4, wherein: (a) the outer shell has its open-ended cylindrical part at the fore end

thereof and the opposite end thereof includes a rocket nozzle, and (b) said inner shell has the fore end thereof forming said restricted opening and its opening is at the aft end thereof.

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6. A filament wound vessel produced by the method claimed in any of Claims 1 to 3.

7. A filament wound vessel substantially

as described and shown in Figure 1 or 2 of the accompanying drawings.

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COMPLETE SPECIFICATION

1 SHEET

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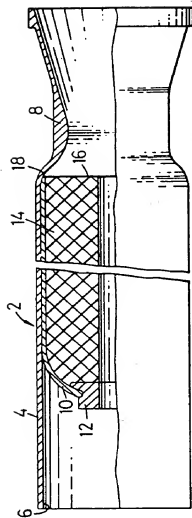


FIG. 1

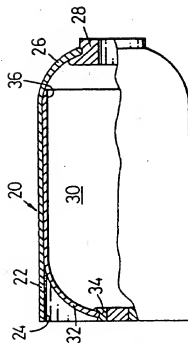


FIG. 2

United States Patent [19]

21 JUN 1982

4,332,631

Herty, III et al.

SCIENCE REFERENCE LIBRARY [45]

Jun. 1, 1982

[54] CASTABLE SILICONE BASED MAGNESIUM
FUELED PROPELLANT[75] Inventors: Charles H. Herty, III; Samuel E.
McClendon, both of Waco, Tex.[73] Assignee: Hercules Incorporated, Wilmington,
Del.

[21] Appl. No.: 164,730

[22] Filed: Jun. 19, 1980

[51] Int. Cl.³ C06B 45/10

[52] U.S. Cl. 149/19.2; 60/207;

60/208; 149/20; 149/114

[58] Field of Search 60/207, 208; 149/19.1,
149/19.2, 19.9, 20, 114

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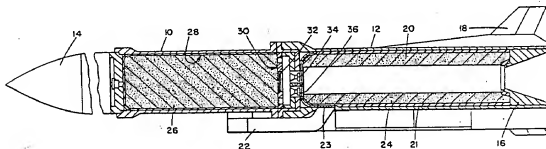
Primary Examiner—Edward A. Miller
 Attorney, Agent, or Firm—Michael B. Keehan

[57]

ABSTRACT

A castable solid propellant composition suitable for use as a fuel generator for a ducted rocket motor is provided. The binder for the composition is a crosslinked polysiloxane. The polysiloxane polymer which is crosslinked to form the binder has a viscosity measured at 77° F. of from about 12 poise to about 50 poise. The metal fuel for the composition is a mixture of spherical and flake magnesium. The composition contains from about 16% to about 20% polysiloxane binder and from 55% to 63% magnesium fuel. The oxidizer is principally ammonium perchlorate.

3 Claims, 2 Drawing Figures



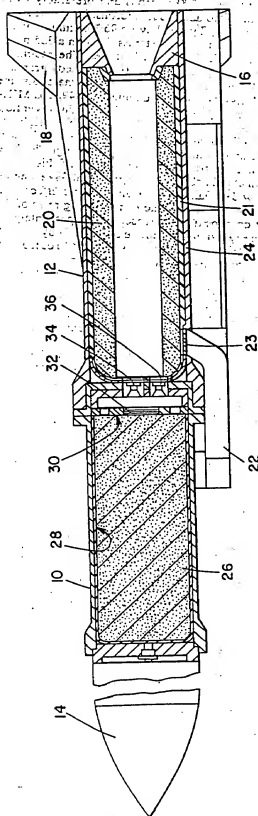


FIG. 1

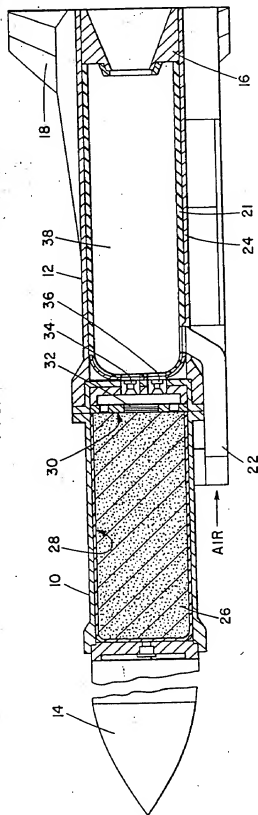


FIG. 2

CASTABLE SILICONE BASED MAGNESIUM FUELED PROPELLANT

The Government has rights in this invention pursuant to contract No. F04611-76-C-0042 awarded by the Department of the Air Force.

This invention relates to a castable silicone based magnesium fueled propellant.

More particularly, this invention relates to a solid propellant fuel generator particularly suitable for use in a ducted rocket motor which solid propellant employs a polysiloxane binder and magnesium as fuel.

BACKGROUND

Air augmented rockets offer potential performance advantages over conventional rocket systems. The advantages inherent in air augmented rockets have long been recognized. The advantages result in using atmospheric air as an oxidizer, supplanting a significant fraction of the oxidizer that must be transported in a more conventional rocket. One member of the air augmented propulsion family of rocket motors is the ducted rocket comprising a solid propellant fuel generator, a booster propellant, a secondary combustion chamber including air inlets, and exhaust nozzle. The secondary combustion chamber typically houses an integral solid propellant booster grain and ejectable nozzle. The ducted rocket has potential of delivering an effective specific impulse in the sustain phase (air augmented combustion phase) 50-60% greater than that of a conventional solid propellant powered rocket. This greater effective specific impulse leads to greater mission capability, e.g., longer range, higher velocity, greater payload, etc.

The purpose of a solid propellant fuel generator for a ducted rocket is to deliver to the secondary combustion chamber a combustible-rich exhaust capable of high heat release when combined with the ram air stream provided by the ducted rocket system. Light metals such as boron, aluminum and magnesium are attractive from a thermochemical performance standpoint as a fuel for a solid propellant fuel generator for use in a ducted rocket at high percentage metal loadings, greater than 50% by weight based on the weight of the propellant. One difficulty with light metal loaded propellants for fuel generators for ducted rockets has been low burning rates.

Castable magnesium-rich propellants based on hydrocarbon binders are difficult to formulate with burning rates much in excess of 0.9 in./sec. at 600 psia because of the char formation which takes place during burning. The char formed partially inhibits combustion of the magnesium and impedes the progress of the flame front during burning. Higher burning rates, i.e., greater than 0.9 in./sec. are possible with magnesium-rich propellants based on hydrocarbon binders when ferrocenes or carboranes are included in the propellant compositions. However, these burning rate adjuncts are expensive and impart adverse aging and migration characteristics in propellants containing such materials.

SUMMARY OF THE INVENTION

In accordance with this invention, a castable, solid propellant suitable for use as a fuel generator for a ducted rocket motor is provided, said propellant comprising by weight from about 50-55% spherical magnesium, from about 5 to about 13% flaked magnesium, the combined weight of the spherical and flake magnesium

being from about 55% to about 63% by weight of the propellant, from about 18% to about 25% of oxidizer of which at least about 80% is ammonium perchlorate, from about 16-20% of a polysiloxane binder, and the weight ratio of oxidizer to polysiloxane binder is from about 0.9/1 to about 1.5/1.

The magnesium employed in the castable propellant of this invention comprises a mixture of spherical magnesium and flake magnesium. The spherical magnesium is atomized magnesium having 98% minimum magnesium content and a nominal particle size of from about 44 microns to about 500 microns and preferably from about 44 microns to about 300 microns. The flake magnesium is Grade A, Type I, having 96% minimum magnesium content and 98% must pass through a 325 mesh screen based on U.S. Standard Sieves. The atomized magnesium employed is Type III (atomized) granulation Nos. 16 and 17. These magnesium materials are defined in JAN-M-382A dated June 23, 1949, and MIL-M-38213. The propellant compositions of this invention contain from about 55% to about 63% by weight of magnesium of which from about 50-55% is spherical magnesium and from about 5-15% by weight is flake magnesium. The propellant compositions of this invention contain at least about 5% by weight of flake magnesium. Below 5% flake magnesium, adverse effects can occur in combustion efficiency and burning rates of the propellant composition decrease. As the percentage of flake magnesium increases above about 15% by weight, the propellant becomes extremely difficult to cast and at 20% by weight the propellant is uncastable.

The solid oxidizer which can be employed in the castable propellant of this invention is preferably ammonium perchlorate having a particle size of from about 1 micron to about 200 microns. At least 80% by weight of the solid oxidizer must be ammonium perchlorate. Other oxidizers which can be employed with ammonium perchlorate in amounts of up to about 20% of the total solid oxidizer include potassium perchlorate, cyclo-trimethylene trinitramine (RDX) and cyclotetramethylene tetranitramine (HMX).

The polymeric binder which can be employed in the castable propellant of this invention is a polysiloxane polymer which is crosslinked at room temperature, i.e., about 22° C. The polysiloxane binder has a polymeric backbone of alternating silicon and oxygen atoms with pendent hydrocarbon groups on the silicon atoms. The pendent hydrocarbon groups are predominantly methyl groups but some phenyl groups, up to about 10% by weight, are often included in the polymer. The polysiloxane polymers which can be employed are low molecular weight polymers having a viscosity measured at 77° F. of from about 12 poises to about 50 poises. As viscosity of the polymer increases above about 50 poises, the castability of the propellant being prepared is adversely effected and casting becomes very difficult. Castability can be improved by the addition of low viscosity (about 50 cps) silicone diluents or plasticizers. The polysiloxane binder employed in the propellant of this invention decomposes without forming an inhibiting char layer around the magnesium particles within the propellant. The weight ratio of oxidizer/binder in the propellant of this invention is from about 0.9 to about 1.5. If the oxidizer/binder ratio exceeds about 1.5, then the percentage of magnesium fuel being oxidized to magnesium oxide and delivered to the ram burner decreases resulting in reduction in energy provided by the rocket motor. If the oxidizer/binder ratio falls

below 0.9, then flame temperature is lowered which results in decreased efficiency in expelling fuel from the generator into the secondary combustion chamber and excessive slagging of the nozzles can occur due to the presence of liquid magnesium.

Polysiloxane binders of the type employed in the fuel generator propellant of this invention are available commercially. Commercial polysiloxane binders that can be employed include, without limitation, General Electric's RTV silicone 602, 615, and 910 and Dow Corning's Sylgard resins 182 and 184. The polysiloxanes are crosslinked to form the polysiloxane binder. Illustrative crosslinking agents include dibutyl tin dilaurate, ethyl silicate and alkyltrialkoxysilane. Plasticizers or silicone diluents can be employed in the binders to reduce viscosity. Typical plasticizers are polydimethyl siloxanes having a viscosity of about 50 centipoises.

A typical ducted rocket motor in which the solid fuel generator propellant of this invention can be employed is illustrated in the drawings in which:

FIG. 1 is a longitudinal view of a ducted rocket motor partly in cross-section and

FIG. 2 is a longitudinal view, partly in cross-section further illustrating a ducted rocket motor containing a solid fuel generator propellant.

As shown in FIG. 1, the ducted rocket motor consists of a forward cylindrical section 10, an aft cylindrical section 12, a nose 14, nozzle 16 and airfoil 18. A booster propellant 20 is housed in aft section 12. An air inlet duct 22 is mounted parallel to the longitudinal axis of the rocket motor adjacent the forward end of the aft section 12 of the rocket motor and mounted contiguous with the exterior surface of the assembled rocket motor. The air inlet duct 22 extends forward of the aft end of the forward section 10. The air inlet passes through an opening in the side wall 24 of the aft section 12 and is attached (not shown) to the side wall 24 forming aft section 12. Booster propellant 20 is cast over the ram burner insulation 21 and port cover 23. Port cover 23 seals the opening in side wall 24 into which air duct 22 connects.

The fuel generator propellant 26 of this invention is stored within chamber 28 of forward section 10. At the base 30 of the fuel generator propellant 26 is an igniter 32. Spaced from base 30 of fuel generator propellant 26 are two nozzles 34, 36. In operation, booster propellant 20 is ignited and burns to provide the initial thrust for the ducted rocket. Upon burn out, the chamber which held the booster propellant 20 becomes the secondary combustion chamber for the fuel generator propellant. Igniter 32, on command, ignites fuel generator propellant 26 and the exhaust gases containing a high percentage of magnesium particles pass through nozzles 34, 36 into secondary combustion chamber 38 (see FIG. 2). Air passes through the air inlet duct 22 and being under a high pressure resulting from the high velocity of the rocket, the air rushes into secondary chamber 38 and mixes with the hot magnesium particles and other combustion gases resulting from burning of the fuel generator propellant. Additional combustion takes place in the secondary combustion chamber 38 and the exhaust gases pass through nozzle 16 providing additional thrust for the rocket motor.

The examples which follow illustrate a process for preparation of the castable fuel generator propellant of this invention and its ballistic properties. In the examples and throughout this specification, percentages are by weight unless otherwise specified.

EXAMPLE 1

A mixer is charged with polysiloxane polymer having a viscosity of 40 poises at 77° F., polydimethyl siloxane plasticizer, iron oxide (combustion catalyst) and about 50% of the total magnesium metal which is to be employed as a fuel in the final propellant composition. Hexane is added to the mixer in a sufficient amount so that a slurry is produced without intensive mixing action. The resulting slurry is mixed for about 5 minutes and all additional magnesium is added to the slurry and mixing is continued for an additional 5 minutes. Ammonium perchlorate is then added to the resulting slurry in two equal increments with mixing continuing for about 15 minutes after addition of each increment. The resulting propellant mixture is then mixed under vacuum at 170° F. until the temperature of the mixture stabilizes at about 170° F. Mixing is continued under vacuum until the vacuum on the propellant mixture is at least 28 inches of mercury (minimum). After the temperature of the propellant mixture has stabilized and the vacuum of the mixture is at least 28 inches of mercury, the temperature of the propellant mass is reduced to about 100° F. The cooled propellant mass is then mixed for an additional 60 minutes under vacuum. A curing agent for the polysiloxane is then added to the propellant mass with mixing continuing for an additional 20 minutes under a vacuum of about 28 inches mercury. The resulting propellant is then cast into molds of desired sizes and shapes.

EXAMPLES 2-4

Following the mixing procedure of Example 1, castable solid propellants for a fuel generator are prepared. The compositions and properties of the propellants are given in Table I below.

In the composition listed in Table I the spherical magnesium employed is Type III, granulation No. 17 (nominal mesh size of 50 to 100 based on U.S. Standard Sieves). The flake magnesium is Type 1 of which 98% passes through a 325 mesh screen (based on U.S. Standard Sieves). The ammonium perchlorate employed in the composition of Examples 2 and 3 has a particle size of about 10 microns. The ammonium perchlorate employed in the composition of Example 4 is a bimodal mixture, about 66% having a particle size of 10 microns and 33% having a particle size of 200 microns.

TABLE I

	Ex. 2	Ex. 3	Ex. 4
Composition, wt %			
Magnesium, Spherical	50	50	55
Magnesium, Flake	13	10	5
Ammonium Perchlorate	20	21	20
Polysiloxane Binder	17	19	19
Polysiloxane, 30 poise ^(a)	9.3	10.4	14.4
Polysiloxane, 30 centipoise ^(b)	6.8	7.6	4.0
Alkyltrialkoxysilane ^(c)	0.9	1.0	1.6
Ballistic Properties			
Theoretical Heating Value, BTU/lb.	9055	9002	9116
Burn Rate at 400 psi, in./sec. (measured)	0.99	0.78	0.33
Burn Rate at 600 psi, in./sec. (calc)	1.16	0.86	0.32
Pressure Exponent (n)	0.42	0.30	0.1
α_p (-65 to 165° F.), %/°F.	0.065	0.11	0.18
α_k (-65 to 165° F.), %/°F.	0.108	0.16	0.19
Theoretical Density, lb./cu.in.			
65 lb. Mix Viscosity, 1 hr. kp	0.057	0.056	0.056
Potlife, hr.	15.8	10.1	9.3
	>8	>8	>8

TABLE I-continued

	Ex. 2	Ex. 3	Ex. 4
Tensile Properties at 77° F.			
cm. % (elongation)	12	14	7
psi (tensile strength)	80	78	45
E. psi (modulus)	951	816	1108
Secondary Combustion			
Expulsion Efficiency, % ^(a)	99+	99+	93-96
Combustion Efficiency, % ^(d)	96-100	97-99	77-99
Hazard Sensitivity			
Autoignition (5 sec.), °F.	—	—	—
Autoignition (5 sec.), °F.	>650	>650	>650
Impact Sensitivity, in.-lb.	43.7	46.0	37.5
Spark Sensitivity, Joules	—	—	—

^(a)RTV-615 available from General Electric Company, (polymer)

^(b)RTV-910 available from General Electric Company, (plasticizer)

^(c)RTV-615B available from General Electric Company, (crosslinker)

^(d)Weight % of original propellant ejected from gas generator

^(e)Percentage of measured ballistic performance compared with theoretical ballistic performance

The burning rates of the castable propellant of this invention at a given burning pressure can be increased by increasing the flake magnesium content of the propellant, and/or by adding a combustion catalyst, such as iron oxide.

What I claim and desire to protect by Letters Patent is:

1. A castable solid propellant composition suitable for use as a fuel generator for a ducted rocket motor, said composition comprising by weight

(a) from about 50% to about 55% of spherical magnesium having a particle size range from about 44 microns to about 500 microns, and from about 5% to about 13% of flake magnesium having a particle size range such that 100% passes through 40 mesh and about 98% passes through 325 mesh U.S. Standard Sieve Series, the combined percentages of spherical magnesium and flake magnesium being from 55% to about 63% by weight of the propellant composition, and

(b) from about 18% to about 25% of solid oxidizer of which at least about 80% is ammonium perchlorate, and

(c) from about 16% to about 20% of a polysiloxane binder comprising a polysiloxane polymer having a viscosity measured at 77° F. of from about 12 poise to about 50 poise, and a cross-linking agent for said polysiloxane polymer,

the weight ratio of solid oxidizer to polysiloxane polymer being from about 0.9/1 to about 1.5/1.

2. The castable solid propellant composition of claim 1 in which the spherical magnesium has a particle size of from about 44 microns to about 300 microns.

3. The castable solid propellant composition of claim 2 in which the solid oxidizer is 100% ammonium perchlorate.

* * * * *